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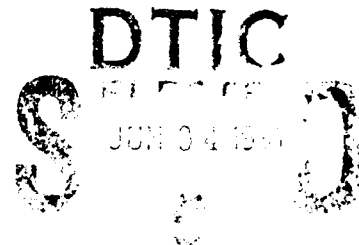
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DEPARTMENT OF DEFENCE

DEFENCE SCIENCE AND TECHNOLOGY ORGANISATION

AERONAUTICAL RESEARCH LABORATORY

MELBOURNE, VICTORIA

Aircraft Structures Technical Memorandum 578

A REVIEW OF AUSTRALIAN AND NEW ZEALAND

INVESTIGATIONS ON AERONAUTICAL FATIGUE DURING THE

PERIOD APRIL 1989 TO MARCH 1991

Edited by

G.S. JOST

Approved for public release

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SUMMARY

This document was prepared for presentation to the 22nd Conference of the International Committee on Aeronautical Fatigue scheduled to be held in Tokyo, Japan, on May 20 and 21, 1991.

A review is given of the aircraft fatigue research and associated activities which form part of the programmes of the Aeronautical Research Laboratory, Universities, the Civil Aviation Authority, the Australian aircraft industry and the Defence Scientific Establishment, New Zealand. The major topics discussed include the fatigue of both civil and military aircraft structures, fatigue damage detection, analysis and repair and fatigue life monitoring and assessment.

DSTO ARL

MELBOURNE

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9.1 INTRODUCTION

This review of Australian investigations on aeronautical fatigue in the 1989 to 1991 biennium comprises the collective inputs from the Australian and New Zealand organisations listed below. The author acknowledges with appreciation the contributions of those shown against each of the items in the Review.

The abbreviations and addresses of contributing organisations are as follows:

ARL Aeronautical Research Laboratory, 506 Lorimer St., Fishermens Bend, Victoria 3207.

ASTA Aerospace Technologies of Australia, PO Box 4, Port Melbourne, Victoria 3207.

CAA Civil Aviation Authority, PO Box 367, Canberra, ACT 2601.

HdHV Hawker de Havilland (Vic) Pty Ltd, GPO Box 779 H, Melbourne, Victoria 3001.

MU Monash University, Clayton, Victoria 3168.

RMIT Royal Melbourne Institute of Technology, GPO Box 2476 V, Melbourne, Victoria 3001.

UoM University of Melbourne, Parkville, Victoria 3052.

DSE Defence Scientific Establishment, Auckland, New Zealand.

9.2 FATIGUE PROGRAMMES ON MILITARY AIRCRAFT

9.2.1 F-111 Wing Pivot Fitting Inspection Intervals (J.M. Grandage – ARL)

In conjunction with the development of the boron/epoxy doubler for the upper plate of the wing pivot fitting [1] the RAAF has requested that ARL estimate in-service inspection intervals for the No. 2 stiffener runout, Fig.1. ARL has obtained interim estimates of inspection intervals for the runout, both with and without doublers. The task is unusual in that the problem area is nominally in compression during positive manoeuvre loads. However the wings are regularly subjected to cold proof load testing, essentially to "inspect" for cracks in the lower load path in the wing and carry-through box, and during proof loading a local secondary bending influence in the upper plate of the wing pivot fitting causes local compressive yielding in the stiffener runouts. A residual tensile stress remains after removal of the proof load and hence all subsequent service loads below about 2g result in tensile cycling.

Crack growth data from service aircraft are available from three wings. In preference to using crack growth prediction methodology, one of these crack growth curves has been used for deriving inspection intervals in RAAF service. To allow for the influence of the doubler on the stress response, strain survey data from five wings, in some cases both with and without doublers, were utilised. The derivation of stress response, and also the residual stress following proof loading, given strain data from the yielded material, was based on data from an experimental stress/strain investigation by General Dynamics. A review of strain data from the five wings showed an excessive between-wing variability in strain response, and this was allowed for by basing the assessment on strain data from that aircraft showing the greatest response. Comparative crack growth calculations with and without the doublers indicated that the doubler increases the inspection interval by a factor of about 2.0. The calculations also suggest that much of the crack growth results from loads in the 0g to 2g range. In view of the fact that such load levels are not explicitly recorded by the fatigue meters fitted to the RAAF fleet, this indicates a need for more comprehensive loads monitoring of individual aircraft.

Interim estimates of inspection intervals have been provided to the RAAF and current work is aimed at reassessing some of the input data to those estimates [2].

9.2.2   F-111 Strain Survey (D.C. Lombardo - ARL)

As part of its support of the Royal Australian Air Force's (RAAF) fleet of F-111 aircraft, ARL has conducted strain surveys on a complete F-111C aircraft. The aircraft, designated A8-113 and based at RAAF Base Amberley (Queensland), is fitted with approximately 130 strain gauges (some from previous and continuing tests and some new).

Two separate surveys were carried out. The first (low-load) survey was at Amberley in April 1990, [3], the reasons being fourfold. First, and most important, transfer functions were required which relate the strains at AFDAS* gauge locations to those at DADTA control points. Secondly, they provided a general base of information relating strains (and hence stresses) to applied loads. Thirdly, they were done to support the boron-epoxy doubler reinforcement scheme. This scheme, developed at ARL, [5], is intended to reinforce the wing pivot fitting, Fig.1, and to minimise cracking at stiffener runout No.2 (Section 9.2.1), and these surveys have allowed the determination of residual stresses produced in the course of reinforcement. Finally, aircraft A8-113 underwent a standard Cold Proof Load Test during which time a complete strain survey to 100% proof load was conducted. The preliminary strain survey could therefore be considered as validation testing for all gauging and instrumentation.

For both preliminary and full-load surveys duplicated down-up load excursions were made. At Amberley, the first survey was made on the clean aircraft. The wing pivot fitting boron-epoxy doublers were then fitted and a second survey was carried out. The residual strains resulting from doubler fitment were also determined at this time.

The proof load strain survey was carried out at Sacramento in September 1990. Three load excursions were made: at ambient temperature, at -40° and again at ambient temperature. This second survey validated the extrapolations made from the first survey extremely closely, and has provided strain per unit load data at nearly 100 locations throughout the airframe. Good use was made of the data from gauges in the upper wing pivot fitting region in carrying out the study reported in the previous section.

9.2.3   DADTA on RAAF Macchi MB 326 H Aircraft (P.C. Garrick - HdHV)

Earlier work on this programme has been reported in previous ICAF National Reviews.

Crack growth predictions for the Macchi have been developed using a theoretical crack growth model. The model has been calibrated against fatigue test results.

Determination of service inspection periods for the fleet are presently being undertaken.

9.2.4   GAF Nomad (L. Tuller - ASTA)

Progress on the Nomad full-scale fatigue test has been reported regularly in past ICAF Reviews. Technical results on specific aspects of the overall programme are published mainly in Project Notes, Certification Reports and Test Specifications.

One of the more significant aspects highlighted by the fatigue testing programme has been the discovery of the fatigue-sensitive nature of the stub-wing [1]. Following the failure of the second stub-wing front

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* AFDAS - Aircraft Fatigue Data Analysis System is the Australian-designed fatigue monitoring system being implemented in all major RAAF aircraft fleets, - see, for example [4].

spar upper cap at the wing strut pick-up fitting position in 1985 after only 37,000 flights (the first stub-wing also failed at this location after 138,000 flights), the stub-wing was rebuilt with a new spar. The opportunity was taken to rework all holes in fatigue-prone regions of the new spar using the cold-expansion process. This spar·has now reached virtual failure after 131,000 flights at the same rivet hole location (B.L.47.6) as had the first two spars. No firm conclusion can be drawn, however, about the effectiveness of cold-expansion in delaying/slowing the propagation of the crack until examination of the fracture surfaces by fractography. Earlier concern about the detrimental effect that the small edge margins of holes in the stub wing upper spar cap might have, do not seem to have been justified.

A detailed strain-survey of the spar on the side opposite the failure location is to be carried out to provide for accurate representation of strains in a coupon fatigue testing programme. This work is aimed at verifying any proposed recommendation for this area in service aircraft.

The overall fatigue test objective included the determination of the critical fatigue failure locations in the main wing. All main wing failures to date have been economically repairable, and are thus unlikely to define the economic life of the wing. Wing loading of increased severity is planned for the remainder of the test, aimed at forcing catastrophic fatigue failure of the main wing soon. The increased loads cannot, however, be supported by the stubwing, and a by-pass structure has been built to react, via shortened struts, the increased flight loads, Fig. 2. Ground loads supplied to the stub-wing will be unaffected. Testing will resume shortly.

A flight testing programme is to be carried out using Nomad aircraft LS-42. This aircraft has been fitted with 120 strain gauge bridges. A new telemetry system has been designed, tested and installed. Wing struts and flap control rods have been subjected to laboratory calibration to establish strain per unit load relationships. The aircraft is presently undergoing ground loading checks, prior to preliminary flight testing [6,7]. The flight test programme has been delayed for the past year when the aircraft was required for urgent investigations relating to a fatal Nomad accident.

9.2.5   F/A-18 full scale fatigue test (A.D. Graham - ARL)

In a collaborative testing programme between the RAAF and the CF (Canada), ARL is developing a full scale test of the aft fuselage and empennage of an F/A-18, in which manoeuvre loads and dynamic buffet loads will be applied simultaneously to an F/A-18 test article [8]. Dynamic modes at 15 Hz, 45 Hz and 90 Hz will be excited in the horizontal and vertical tails of the structure, simulating the in-service buffet environment present at high angles of attack. Acceleration levels of up to 600g at the tips of the fins will be generated.

ARL has developed a novel loading technique employing soft air springs which enables the manoeuvre loads to be applied without significantly affecting the dynamic response of the structure. An array of pneumatic air bag pairs controlled by an ARL-designed servo-valve will enable manoeuvre loads to be applied at near in-service loading rates. The technique has been successfully demonstrated in a test rig using an F/A-18 horizontal stabilator. A loading configuration utilizing one air bag pair is shown in Fig.3. Each pair of air bags will be capable of applying 2250 kg in either direction.

A full scale test rig is being assembled to allow for a five-channel pneumatic array on each horizontal stabilator and a ten-channel pneumatic array on each vertical fin, Fig.4. The fuselage and engine inertia loads to be applied to the test article will also employ the pneumatic loading system.

The dynamic loads will be applied to the structure using high stroke, high load capacity electromagnetic shakers. Six shakers with a peak-to-peak displacement of 100 mm, a load capacity of 19.6 kN for 100 ms

and a maximum velocity of 5 m/s are being manufactured by LING USA specifically for the test. Two shakers will be used on each vertical fin, with one on each stabilator.

The loading systems and control strategies are being developed on an F/A-18 centre/aft fuselage on loan from the US Navy; formerly the static test article STO1 used by MCAIR for demonstration of design loads. A dummy forward fuselage has been spliced onto STO1 and the test article has been suspended in the main test frame at the wing attachment points by two vertical and one lateral horizontal hydraulic actuator. The test article is pivoted in a spherical bearing attached to the dummy nose.

To ensure that the shaker attachment and air bag loading directions remain sensibly normal to the surfaces being loaded, the bulk displacement of the aft fuselage caused by manoeuvre loading will be minimized by employing an active reaction system using the hydraulic actuators suspending the test article in conjunction with a displacement controlled servo system.

To ensure that the correct loads are induced into the engine mounts and associated supporting structure, engines will be installed in the test article, and thrust loading applied. Dummy F404 engines have been made to represent the stiffness and inertial properties of real engines. These have been used during development testing. However, ARL has just received two unserviceable engines from the US Navy which have been provided for the test programme. These will initially be used to help fine tune the dummy engines, which will be used until the real engines are modified to allow thrust and inertia loading systems to be attached.

The test rig control system is under development and significant effort is being given to developing the test spectra and loads. Two high frequency uniaxial fatigue testing machines are being purchased from MTS to assist in the support coupon testing program. The current prediction date for test start-up is mid-1993.

9.2.6 F/A-18 LAU-7A Launcher Guide Rail Fatigue (D.S. Saunders - ARL)

Over the past few years a fatigue problem with the LAU-7A launcher on the RAAF F/A-18 has been observed. It is likely that the problem has arisen because of the wing tip mounting of the LAU-7A launcher and the fact that the AIM-9 missiles (or dummies) are usually flown on the aircraft. The fatigue cracking has led, in some cases, to overload failure of the guide rail thus rendering them inoperable.

ARL has conducted a fatigue testing programme to determine what loads are causing the fatigue cracking. The scope of the work covered inertia loads, and it was found that by factoring up the N_z (centre-of-gravity) loads, service failure fracture surfaces could be reproduced. The testing showed that there were many inertia load levels which would result in fatigue crack growth. The particular spectrum used was considered to be a reasonable simulation of the inertia loads acting on the missile and thus provided the spectrum which was used in the testing of a repair for the launcher guide rail.

A simple, mechanically fastened patch has been designed. The installation of the patch requires some machining of the cracked guide rail and, on installation, the patch itself forms part of the guide rail. To demonstrate the feasibility of the proposed repair, patches machined from alloy steel were tested initially. However, to decrease the weight of the repair, and to achieve some degree of compatibility of materials, an aluminium alloy patch is now being developed. This aluminium alloy patch meets the limit load requirements for the worst-case missile loading condition. Fatigue testing of the patch using the spectrum described above is still continuing.

The mechanical patches have been designed for ease of replacement and inspection and are relatively inexpensive components to manufacture. The present design of patch, if proven successful under flight

conditions, will allow refurbishment of all fatigue cracked launcher guide rails so that they become fully functional.

9.2.7   F/A-18 Bulkhead Fatigue Test (I. Anderson, G.W. Revill - ARL)

The first 32 Australian F/A-18 aircraft were delivered with an early configuration fuselage station 488 bulkhead, which is one of the three wing support bulkheads. The mould-line flanges on these bulkheads (F143 configured) are such that their fatigue life may be inadequate. It is intended that these early configuration bulkheads be modified to increase their fatigue life. The modification involves re-shaping the mould-line flange from the F143 configuration to a 6" radius configuration, Fig.5. Tests of free standing bulkheads performed by McDonnell Douglas and Northrop have shown that the 6" radius configuration has an adequate fatigue life. Bulkheads with the F143 flange profile have not been freestand fatigue tested.

ARL has conducted a fatigue test on a free standing F/A-18 fuselage station 488 bulkhead. Details of the test are provided in [9]. The test was similar to those performed by McDonnell Douglas Aircraft Company and Northrop Corporation. The spectrum applied to the bulkhead was based on the full-scale fatigue test load spectrum for the F/A-18 and was the same as that applied to the bulkheads tested by McDonnell Douglas and Northrop. For the first 7500 simulated flight hours of testing the mould-line flange area of the bulkhead was in the F143 configuration. At the conclusion of 7500 hours, no cracks were detected within the test area. The mould-line flange area of the bulkhead was then modified to the 6" radius configuration, and testing was resumed to failure at 16,200 hours. The bulkhead failed from a fatigue crack in the forward inner edge of the mould-line flange. Detailed inspection of the fracture surface and the un-failed side of the bulkhead revealed a number of other cracks in the aft mould-line flange and in the centre-line of the bulkhead. The conclusion drawn from the test was that the first 32 Australian F/A-18 aircraft delivered with F143 configured mould-line flanges have adequate fatigue life up to the proposed time of rework.

It is proposed to repeat the bulkhead test to a spectrum of loads more representative of Australian usage. It is anticipated that this test will be performed during 1992.

9.2.8   Fatigue Monitoring of RAAF F/A-18 Aircraft (D. McIlroy - HdHV)

The F/A-18 Maintenance Data and Service Life Monitoring System has now acquired operational capability on both the airframe (Airframe Service Life Monitoring Program) and engine (Engine Condition Assessment Program) sub-systems.

ASLMP reports are generated quarterly, covering fleet operations and categorising data by fleet, squadron, type of flying, etc. A recent extension to the ASLMP report compares the wing-root fatigue index to the usage-index derived from unit-level analysis (Mission Severity Monitoring Program). Fatigue consumption continues to be monitored closely.

ECAP reporting has been conducted monthly since September 1990, with concurrent updates of ECAP data to the RAAF's Computer-Aided Maintenance Management database.

Work continues on an allied project for the Aircraft Structures Division, ARL. Software development is being completed against recently extended requirements, which will be used to process all RAAF Hornet flights. Data analysis will aid determination of the loading spectrum for the aft fuselage fatigue test (Section 9.2.5).

9.2.9   F-404 Engine Life Usage (N.S. Swansson - ARL)

A comprehensive review of low cycle fatigue usage of the F-404 engine fitted to F/A-18 fighter aircraft has recently been completed [10-14]. This is the first engine used by the Australian defence forces for which fleet-wide usage measurement and flight history data are available.

Mainly because the average flight duration is shorter, the rate of fatigue life usage in Australia is more severe than for typical overseas operators. Usage severity for various mission types (Types of Flying) was characterised. The Life Usage Indices adopted by the manufacturer, which employ fairly simple equivalent life algorithms, compared well with ARL estimates which employed a relative life methodology with data from alternative sources. Representative flight history data supplied to the manufacturer for design analysis, enabled fine tuning of the usage algorithms and led to a useful increase in life for some components.

9.3 FATIGUE IN CIVIL AIRCRAFT

9.3.1   Fuselage Lap Joint Cracking in Fokker F28 Aircraft (R.B. Douglas - CAA)

Since the spectacular rupture of the Aloha Airlines Boeing 737 in April 1988, the perplexing problem of multi-site damage in pressurised fuselages has received special attention from aircraft manufacturers, airlines, and airworthiness authorities. Despite these efforts, new and unanticipated problems still occur.

The most recent of these problems to occur in Australia is cracking in the lap joints just above the fuselage floor line (at Stringers 17 and 58), behind the wing (between Frames 13345 and 14285) as shown in Fig.6, in several Fokker F28 aircraft. In this area the 0.040 inch thick skins are joined by two rows of dimpled countersunk rivets. An initial fortuitous discovery, followed up by a fleet-wide inspection, has revealed multiple cracks around the dimples in the outer skin at the upper row of rivets, and through the rivet holes in the inner skin at the lower row of rivets. The affected group of aircraft had between 50,000 and 55,000 pressurised flight cycles, and in several instances individual cracks had linked up over several rivets, the longest such crack being 8 inches. Initial fractography indicates that cracking has been present for up to 17,000 cycles; this is still to be confirmed.

The original Fokker inspection program was not suitable for detecting all the cracks reliably, so an impedance-plane low-frequency eddy current gliding probe supplementary technique has been developed by an Australian airline. Fokker have suggested that they will incorporate this technique into a future Service Bulletin.

9.3.2   Supplemental Structural Inspection Programme for NORD 298 Aircraft (R.B. Douglas - CAA)

As well as pressurised fuselages, the Aloha accident has heightened the interest in ageing airframes generally. Manufacturers of large aircraft - greater than 75,000 pounds MTOW - had by 1988 already made considerable progress toward the development of Supplemental Structural Inspection Documents (known as SSIDs). These extra inspections were the final product of complete re-analyses of the structural integrity of older aircraft types in the light of the latest damage tolerance rules - a massive engineering effort.

Following the Aloha accident, existing SSIDs have been re-examined, and the extension of SSIDs to cover smaller commuter aircraft have been considered. There are several problems with SSIDs for commuter aircraft, however. One is that the engineering expertise and sheer effort required may be beyond the

capability of some of the smaller manufacturers. Another is that even if technically capable, some may not be in a position to be able to divert so much of their scarce resources to the support of ageing aircraft that are no longer in production.

One Australian commuter airline took the bold step a couple of years ago to engage a consultant aeronautical engineer to develop a SSID for their ageing fleet of Nord 298 Mohawk aircraft, Fig. 7. Their aircraft were amongst the oldest in the world fleet, and no-one had yet developed a SSID for the type. The Civil Aviation Authority has only recently received the completed Nord 298 SSID from the consultant.

The initial assessment is that the Nord 298 SSID, also applicable to the standard Nord 262, is a first class effort, the equal of those which have been produced by major overseas manufacturers. Such an ambitious project could not have been accomplished without the excellent spirit of co-operation that developed between the consultant, the airline, the original manufacturer Aerospatiàle, and the Civil Aviation Authority.

The U.S. FAA has been advised, with the recommendation that the Nord 298 SSID be considered for mandatory implementation as part of the FAA's Aging Commuter Airplane review. The Nord 298 SSID meets the latest U.S. damage tolerance rules.
Implementation of the SSID should enable the Nord 298 aircraft to safely remain in service for the foreseeable future. The SSID will be regularly reviewed by the consultant, and amended as necessary to incorporate new NDI developments and the effects of defects found on both local and foreign fleets.

The engineering analysis confirmed the general robustness of the 30-year-old design, but at the same time highlighted several critical areas where damage could grow to potentially catastrophic proportions and not be found by the normal maintenance program. Some defects of this type, including serious wing main spar corrosion inside the engine nacelle, and mis-drilled holes in the main fuselage frames, were uncovered as part of the trial SSID inspections.

9.3.3   Main Rotor Blade Failure in Robinson R-22 Helicopter (R.B. Douglas - CAA)

In May 1990 a fatal accident occurred to a Robinson R-22 helicopter in Australia as a result of an in-flight fatigue failure of one main rotor blade. The component history records were incomplete but it was established that the blade had achieved about 2257 hours time in service. The promulgated mandatory fatigue retirement life is 2000 hours.

Failure occurred in the 7075 aluminium alloy root fitting, Fig. 7, at the most inboard threaded bolt hole of 11 bolts which fasten the stainless steel leading edge D-spar to the fitting. The failure location is shown in Fig. 8. and the fracture surface in Fig. 9.

The laboratory investigation [15] initially focused on the thread damage visible in the bolt hole; however this is a known problem resulting from chemical etching during the anodising process and was established as not being a primary cause of the failure.

The bolted joint had been designed and developed to optimise the load transfer for maximum fatigue performance. In addition to the bolts, the spar is bonded to the fitting to eliminate fretting. Laboratory examination disclosed some anomalies which indicated a lack of clamp-up during bond cure. This in turn suggested that the inboard bolt was not properly torqued prior to bonding. It was eventually established that there was some positional mismatch between the holes in the spar and the fitting, sufficient to cause interference between the bolt shank and the opposite sides of the holes in

the spar and the fitting. Contact marks on the bolt and in the holes confirms this. The contention is that the resultant frictional forces were sufficient to absorb much of the specified bolt tightening torque, resulting in inadequate clamping at this location. Further, the orientation of the mismatch was such that there was significant preload contact between the bolt and the inboard side of the hole in the spar, and the outboard side of the hole in the fitting, resulting in substantially increased load transfer at this first bolt location, Fig. 10.

Airworthiness control has been achieved by instituting an inspection program calling for bolt removal and eddy current inspection of the hole for cracking at 1500 hours time in service, repeating at 200-hour intervals until retirement, based on measured crack growth, and a check for the hole mismatch condition. The manufacturer has instituted additional quality control procedures to ensure conformity.

As at January 1991, no further blades have been found cracked, nor any with the hole mismatch condition.

9.3.4 Janus Glider Wing Fatigue Test (C.A. Patching, L.A. Wood - RMIT)

Progress on this test has been reported in previous ICAF Reviews. The test specimen comprises a complete tip-to-tip wing assembly. The starboard wing was badly damaged in a major accident, and has been fully repaired using a variety of techniques. The port wing was purchased new from the factory.

The full scale fatigue test is continuing and, as at February 1991, over 18,000 equivalent flying hours had been accumulated. There has been substantial growth of minor damage from unrepaired wing sections and several "field repairs". The unrepaired damage was retained to simulate undetected damage of the type that occurs in the field, for example, cracking of the inner surface of the wing skin.

There has been no significant change to the measured strain/g values throughout the duration of the test. Some 15 gauges have become unserviceable through problems associated with installation. As yet there have been no gauge fatigue failures in the foil grid of the gauges. Bending and torsional stiffness have also shown no evidence of changes for either wing.

Interim results of the test programme are listed below for the repaired wing:

1. Delamination of upper skin from unrepaired damage. An unrepaired crack initiated delamination between the inner and outer surfaces on the upper skin, and grew rapidly in the first 600 hours. It was repaired at 613 hours.

2. Delamination of lower skin near the rear spar. A small area of skin delamination extending to the rear spar resulted in skin separation from the rear spar in first 600 hours. It was repaired using insert panels.

3. Unrepaired spanwise delamination on lower surface. Throughout the test there has been only marginal growth at the defect's outer spanwise end, indicating that the tension surface is not readily susceptible to damage.

4. Deliberate damage to upper spar boom. Deliberate damage was introduced to the upper spar boom to simulate impact with a sharp object. A sharp-edged steel tool was used to generate a notch shape depression on the spar cap. Standard repairs were made by replacement of a wedge-shaped

segment of material around the notch. There has been no evidence of fatigue damage occurring at the repair.

5. Movement of rear bearing in the root rib. Radial cracks from the bearing were detected at 11,138 hours. At 13,040 hours delamination occurred in that area, and at 13,862 hours the test was stopped and a repair made to the damaged root rib. At 15,098 hours cracking was again detected, necessitating further repairs. No further damage has been detected.

6. Delamination of lower skin near aileron fitting. Separation of the outer surface from the foam core occurred at an aileron push rod guide which is attached to the inner surface of the lower skin. The damage was repaired at about 500 hours by injecting resin. No further damage has occurred.

7. New crack on upper skin. This new chordwise crack was detected on the inner surface of the upper skin at 5,572 hours. At the time of detection it was 130 mm in length, and it was left unrepaired to monitor its propagation rate. After 15,500 hours its length was 196 mm, a growth rate of about 6.6 mm per 1,000 hours.

8. Propagation of existing gel coat cracks. Considerable prior stress-induced gel coat cracking had occurred on the salvaged wing, but there has been no propagation of these cracks throughout the test.

9. Movement of front bearing in root rib. Radial cracking was detected at about 18,000 hours, and there was movement between the housing and the root rib. As well as repairing the local structure, a new bearing and housing were installed as the bearing showed evidence of restricted movement in one direction.

For the new wing, cracking of the gel coat has occurred on the lower surface. A crack in the gel coat on the tension surface over the lower spar boom was detected at 11,138 hours, 46 mm in length and oriented chordwise. It has not grown in length since discovery, and there is no evidence that the crack has extended beyond the depth of gel coat (nominally 0.4 mm).

Other related work in progress includes the development of improved specimens for static tension and compression testing and fatigue testing; a finite element analysis of the wing root attachment with a view to improving durability; and vibration monitoring of the wing structure throughout the fatigue test.

9.3.5 MDX Helicopter (P.J. Foden - HdHV)

HdH is a major partner with McDonnell Douglas Helicopters in the design of a new eight-place civil utility helicopter. HdH has the responsibility for the design and manufacture of the fuselage. This is of sheet-metal/carbon fibre hybrid construction where the composites are used mainly as thin sheets with either moulded stiffeners or sandwich construction.

Several design-specific programmes are planned, and investigations into the fatigue behaviour of conventional discrete fasteners and of shear panels in the post-buckling state have already been completed.

9.4 FATIGUE DAMAGED STRUCTURE: ANALYSIS, DETECTION AND REPAIR

9.4.1 Stress and Fatigue Analysis of Coldworked Holes (G. Clark, G.S. Jost - ARL)

A model has been developed [16] for predicting residual stresses and crack growth in residual stress fields, and an application of the model to crack growth from cold worked fastener holes in thick section aircraft components has been made. Comparison with experimental results demonstrates that the model can provide useful predictions of critical crack length, and a capability for correctly predicting the maxima and minima in the crack growth rate for cracks from cold-expanded holes. It also permits the observed asymmetry in cracking from cold-worked fastener holes to be better understood.

In the previous Review, brief mention was made of the evaluation of closed-form analytical expressions for the stresses and strains associated with the coldworking of a hole in an elastic-perfectly plastic annulus under plane strain conditions [17]. That work has now been extended to include remote loading, interference fitting and both of these loadings acting together [18]. It is found that the most benign stress state from a fatigue viewpoint is that obtained from interference fitting a cold-expanded hole.

9.4.2 Life Improvement by Cold Expansion of Holes and Interference-fit Fasteners (J.M. Finney - ARL)

Previous work at ARL had indicated that, although cold expansion of open holes in metal specimens greatly improved fatigue life, expansion of a three layer stack (specimen plus two side plates) giving a low-load-transfer joint gave a much smaller life improvement. This degradation appeared to result from fretting at the out-of-plane protrusions of the faying surfaces arising from the cold expansion.

't was thought that the use of interference-fit fasteners after cold expansion might clamp the joint more rigidly, allow less movement for fretting and recover the open-hole life improvement. An experimental programme is underway using all combinations of cold expansions of zero and 4% and fasteners with interference fits of zero, 0.5% and 1.5% in two-fastener three-layer stack-up specimens. The fighter load spectrum used gives a baseline fatigue life of about 2500 flying hours. Although not complete, the results to date indicate that all combinations give roughly the same life improvement, a factor of about six, over the non-cold-expanded neat-fit fastener condition.

The tests have demonstrated that life improvements may have inherent limits in low-load-transfer joints. Initially, the life enhanced specimens failed in the fillet region joining the test section to the gripping area. This was overcome by shot peening this region and many failures then commenced from fretting either at the out-of-plane protrusions from cold expansion, or from regions near the edges of the side plates. An extension of the programme is being planned to examine life improvements in 100% load-transfer joints for which it is anticipated that side-plate fretting will not be such a problem.

9.4.3 An Elastic-Plastic Constitutive Relation for Multi-Axial Cyclic Material Response
(J.F. Williams - UoM, R. Jones - ARL)

Classical plasticity and creep models currently used in most finite element codes today are incapable of accurately modelling the multi-axial, cyclic plasticity response of common engineering materials, particularly at high and low temperatures. In particular, the absence of bulk stress dependency terms renders them incapable of accounting for modern thermo-mechanical response characteristics.

The aim of this project is to further develop the constitutive model of Stouffer and Bodner [19,20] into a specific theoretical model for computer applications and computer controlled testing, supported by experimental evaluation for the prediction of the monotonic and cyclic response of metals including strain

and cyclic hardening or softening under multi-axial stressing. The theory requires data from a set of no more than three or four monotonic tensile tests at differing strain rates, a strain-controlled cyclic fatigue test and three or four creep tests at differing stress levels.

A major advantage in using this approach is that the state variable constitutive equations are (i) more accurate for use in non-linear finite element codes and (ii) easily relatable to observed material response. It is proposed to incorporate this approach into a 3-D elasto-plastic finite element code [21], thus rendering the solution of elasto-plastic problems relatively straightforward.

To date, some success has been achieved from investigations on 7050-T7451 aluminium alloy of which major components of the F/A-18 Hornet aircraft are made [22]. By way of example, it has been found that, under strain controlled loading, a strain hold in the plastic region results in significant stress relaxation which asymptotes to a constant stress level, Fig. 11. This phenomenon is predicted by the Stouffer-Bodner theory, Fig. 12. Similar stress relaxation behaviour occurs at strain holds during fatigue cycling.

Further details of this study are given in [23] and [24].

9.4.4 Development of FRAN Fatigue Reliability Program (D.G. Ford - ARL)

The initial development of FRAN (Fatigue and Reliability ANalysis) has now been completed. This is intended for interactive use in structural systems with programmed models for multi-load response and arbitrary roles for state changes and crack growth at designated hot spots. The "state" of local material defines the probability of crack initiation but damage tolerance analyses are also an option.

FRAN's distinctive features are:

(a)   Systematic sampling biased towards extreme values of random parameters. This corresponds to Monte-Carlo importance sampling and the computations simulate fatigue testing of each structure in a sample.

(b)   Dynamic following of interacting fatigue cracks and initiations follows from solving differential equations for state and crack size in each structure. At some or all locations, cracks may pre-exist, either randomly or fixed in size. At one particular location not all structures need contain initial cracks.

(c)   Standard fracture mechanics, state and retardation rules which may vary in different places or be replaced by user-supplied rules.

(d)   Multistep starting of the ordinary differential equations for crack growth and/or state change including restarts after inspections or repairs as described by user programs.

(e)   User-defined choice of several independent random external load histories and equi-weighted Gauss-type quadrature for derivatives in (b) and (d). These are formed as short load sequences with range-pairs constructed from special points in stress space akin to Gauss points.

(f)   To reduce storage, and to assist later transfer to mini and personal computers FRAN has been written in Pascal.

It often happens that fatigue or reliability predictions must be repeated. For this reason FRAN automatically logs updated input data on separate files and creates and updates a log file. It can also accept load sequences from a file and separate load peaks can be interspersed in time. All these files are in ASCII format for direct editing and may be extended from within the program.

FRAN assumes that fatigue at each location follows separate stages of initiation and crack growth and the progress of these as well as overall risk depends upon all load histories, including phase differences, and the current vector of crack sizes. The model for stresses summarises the general stress analysis which must therefore be programmed by the user.

In FRAN a linked list of structural records is created using Lehmer's counting procedure. A list of parameter or crack records is then attached to each structure and a univariate probability and its quantile is assigned to each parameter, dimension, crack size or critical state.

FRAN is believed to be the first program to allow dynamic development of interactive fatigue processes so that the reliability estimate can incorporate a genuine prediction of structural behaviour. It allows for practical events such as pre-existing cracks and sudden large partial failures. This may expose weaknesses in supporting technologies and there are obvious tradeoffs here between complexity, auxiliary stressing, time or cost and the realism of simulation. Simple models can capture significant effects but general problems require more data in order to be treated accurately. Many features of FRAN can be applied to Monte Carlo simulation.

The relative merits of FRAN and Monte Carlo procedures are discussed in [25].

9.4.5 Determining stress components from thermoelasticity (T.G. Ryall - ARL)

If a body is subjected to an applied dynamic load it is well known that the temperature variation due to the loading is proportional to the sum of the principal stresses [26]. Experimental and theoretical work at ARL has shown that there is also a small, almost parasitic effect, involving a quadratic form of the principal stresses [27]. For a complete description of the stress state it is necessary to obtain the stress tensor rather than merely a single stress invariant; this is particularly true in regions where the non-zero principal stresses have opposite signs. The motivation for obtaining the stress tensor is thus obvious and requires no further comment.

It was first shown in [28] that the sum of principal stresses plus knowledge of the free boundaries was sufficient to determine (in two dimensions) the stress tensor. The particular problem studied in detail was a strip of metal loaded in an unknown manner at the two ends with the lateral sides remaining free. It is not difficult to show the uniqueness property for a completely general problem in two dimensions.

The next step forward [29] came from examining the problem of determining stress components from the sum of the principal stresses in a three-dimensional case. The problem considered here was a prism rigidly restrained at one end and subjected to arbitrary loads at the other. The solution to the problem was shown to be unique provided the cross section was not circular. The uniqueness problem in 3-D is obviously a complex question involving geometry and boundary conditions.

Reverting to 2-D problems, the next stage came from examining and taking advantage of the mean stress effect. A thin plate with a hole was loaded both dynamically and statically; using only the response at the fundamental frequency, the static and dynamic stress tensors were obtained from the experimental (non-simulated) data. A robust non-linear least squares method was applied and the motion of the

specimen was modelled [30]. Known boundary conditions were applied. This particular piece of work is important since it sets forth a methodology for determining residual stresses.

Most recent work [31] deals with the problem of determining stress components from both the first and second harmonic response without using boundary conditions. Since the second harmonic response is small compared to the level of noise, the question of experimental design both spatially and temporally is examined. Simulations have shown that, in the presence of realistic noise, the stress invariants and the principal stresses can be recovered extremely accurately. Stress components can also be determined to a lesser, but nevertheless satisfactory, degree of accuracy, Figs. 13, 14 and 15.

9.4.6 Thermoelastic stress measurement (S.A. Dunn - UoM)

In the previous Review, the potential for using the mean stress dependence of the thermoelastic parameter to measure residual stress was discussed. Results from [32] were presented where it was shown how, for a simple geometry and a uniform applied stress field, the residual stresses can be successfully measured. One of the aims of current work is to extend this technique such that residual stresses for more complex geometries and applied stress distributions may be measured. The mean stress effect arises due to the non-linearity in the temperature response to applied load. It was thought that analysis of the linear and non-linear components of the temperature response could be used to determine the residual stress at any given point. Experiments, however, suggested that there was an additional non-linear component which had not previously been observed. Subsequent analysis showed that this additional non-linearity arose due to the observed movement of the specimen whilst undergoing a change in load. If there is any change in the bulk stresses in the direction in which the specimen is moving at the point of interest, an apparent additional non-linearity results. Without any a-*priori* knowledge of the applied stresses in the region of interest, this additional non-linearity cannot be quantified from a point measurement. This method of residual stress measurement is, therefore, impractical as a point-stress measurement tool. Because of this, measurements over an area and subsequent spatial analysis will have to be carried out to determine residual stresses. In [33] it has been shown how such spatial analysis can be used to determine the mean stresses around a hole. Work yet to be carried out involves extending this work such that residual stress, and not just mean stresses, may be determined.

Another aspect of this work involves the analysis of TERSA (Thermal Evaluation for Residual Stress Assessment). TERSA is based on work carried out by researchers at the Admiralty Research Establishment (UK) and is described in [34] and [35]. In [35], results are presented which suggest that the equilibrium temperature achieved by a specimen being heated at a point by a laser is dependent upon the state of stress at that point. In [34] it was claimed that this effect was due to the stress dependence of specific heat. It has subsequently been shown in [36] that, based on theoretical and experimental evidence, the stress dependence of specific heat is not sufficient to explain the results presented in [35]. Work has been carried out using Angstrom's method for measuring thermal diffusivity at uniaxial stresses of 0MPa and 500 MPa for 6Al-4V titanium. This work showed no significant difference in thermal diffusivity for the two stress levels suggesting that the stress dependence of thermal diffusivity is not sufficient to explain the results presented in [35]. The work remaining in this field involves experiments which are currently underway to repeat some of those presented in [35].

It was shown in [37] that the observed thermoelastic temperature response for some composite laminate configurations is highly frequency dependent. The reason for this was due to the different thermoelastic heat which can be generated in each ply. The heat generated is dependent upon the bulk stress, and therefore fibre orientation, in each ply. These different heats generated in each ply give

rise to significant thermal conduction in the through-thickness direction of a laminate and hence the frequency dependence of the observed surface temperatures (at low frequencies significant conduction occurs whereas at high frequencies there is little time for conduction to occur). In [38] this was modelled using a one-dimensional finite-difference scheme. It was also shown there how this effect has the potential to determine the individual strain components for a composite laminate exhibiting this effect. This problem was also modelled in [39] using an analytical technique applied to investigate the results found when thermally dissimilar materials are bonded together. This enabled the effects of a layer of epoxy on the surface of a composite laminate to be investigated. The analytical solution presented in [39] makes the method of stress separation presented in [38] a more practical proposition. The work remaining in this field involves the bringing together of these two studies to demonstrate practically the viability of the technique in [38] to separate strain components in composite laminates.

9.4.7 Whole Field Deformation and Damage (K.C. Watters - ARL)

Recent developments by Aircraft Structures Division of ARL in the measurement of whole field deformations and stresses and in the measurement of damage are reviewed in [40]. In-plane Moire, shadow Moire, holographic interferometry and microgridding have been used to measure whole field surface deformation of specimens. The thermoelastic method (SPATE) has been used to measure whole field bulk stresses (including residual stresses) on the surface of specimens and components. Current developments using SPATE involve analytical methods to separate the components of the bulk stress and the finite element method to determine the subsurface stress distribution from that measured on the surface. For in-plane Moire, holographic interferometry and microgridding techniques, current developments are aimed at automating data acquisition and processing using video cameras and microcomputers. Techniques based on the direct-current potential-difference method for cracks in metal specimens and the ultrasonic C-scan method for delaminations in composites have been used to measure damage.

9.4.8 Fatigue of 7050 aluminium alloy thick plate (J.M. Finney - ARL)

Manufacturers of thick rolled plate have found it difficult to eliminate porosity from central interior regions and the fear is that such porosity could reduce fatigue life. Unnotched specimens, 6.35 mm thick, were made from surface and interior regions of a 144 mm thick plate of 7050-T7451 aluminium alloy and tested under different fighter load spectra [41]. There was no significant through-thickness effect on fatigue life.

Specimens for these tests were polished to P120C abrasive paper using tap water and it was noticed that multiple initiation sites appeared on the flat surfaces of the specimens. Figure 16 indicates that these arose from intergranular corrosion, probably from the tap water. Polishing with de-ionised water reduced the number of these corrosion areas but the interesting facts are that multiple site cracking still occurred extensively on plain surfaces and the fatigue life was unaltered.

9.4.9 Delta-K-Effective formulae for crack growth (J.M. Finney - ARL)

The concept that fatigue crack growth rate should depend only on the range of stress intensity for which the crack is physically open has the potential to account for the effect of mean stress. Since Elber's first proposal in 1971 that the effective stress range ratio, U, can be defined as

$$U = (S_{max} - S_{op}) / (S_{max} - S_{min})$$

and that for 2024-T3 aluminium alloy

$$U = 0.5 + 0.4R$$

other equations have appeared in the literature, each in turn becoming more complex. Each author demonstrates the superior correlating power of his formulation by reference to one or more sets of da/dN-ΔK-R data. These data are transformed into sets of da/dN-ΔK_{eff} via the several formulae available, and visual graphical comparisons and/or statistical analyses from selected regions of the data demonstrate correlating power. None of these authors has examined the robustness of their conclusions, which is surprising in light of the known variability in crack growth rate data.

Some of the literature equations for ΔK_{eff} were applied to a set of da/dN-ΔK-R data which is shown in Fig. 17. The results are given in Fig. 18; by visual observation and statistical analysis the equation denoted Schijve 1 is superior to the others in correlating ability. This result is not robust since changes to growth rates, (a factor of 1.35 for R = −0.7 and 1/1.35 for R = 0.2 : small changes relative to scatter), make the de Koning equation the best correlator. It follows that any one of the four sets of equations examined would be as satisfactory as any other [42].

To be certain of the superiority of any set of equations, da/dN − ΔK data covering many replications, many materials, a wide range of R-values, and a wide range of growth rates must be examined with statistical techniques that include all data points. The data set examined do not meet this standard, nor do the data used in the published justifications of the various equations examined.

The equations examined for ΔK_{eff} all amount to a horizontal shift in ΔK on a logarithmic scale, the value of which is constant for any R-value and equation set. Given that the shapes of the da/dN-ΔK-R curves in Fig. 17 are not identical nor are better founded curves, it is clear that equations of the type examined do not have the correct functional dependencies to properly correlate the R-effect in crack growth rate.

9.4.10 Stable Tearing in Near-Plane Strain Conditions (G. Clark - ARL)

Over many years, ARL staff have observed significant areas of stable tearing (i.e. microvoid coalescence fracture modes) which occur on fatigue fracture surfaces of high-strength steels and aluminium alloys from aircraft parts and in laboratory specimens. In many instances, the constraint condition has been close to plane strain, and this had led to concern that the tearing fracture mode indicated incipient failure, particularly where such tear bands are visible at an early stage of crack development. This concern was based on assuming that plane strain conditions were associated almost exclusively with unstable fracture. However, consideration of the situations in which tearing has been observed suggests that *stable* tearing is quite common even in situations where plane strain conditions predominate; the tearing can be extensive - in some examples, stable tearing covers much of the fracture surface. Crack shapes after a stable tear may be very distorted compared to those normally expected in fatigue, with large regions of "tunnelling" i.e. crack advance which is stabilised by near-surface ligaments. These observations demonstrate that a single load cycle in fatigue is capable of causing a very large increment in crack length, an observation with significant implications with respect to NDI, the development of fracture control approaches based on fatigue crack growth, and fracture criticality. This stable tearing behaviour may be represented by reference to the R-curve approach to fracture; while the traditional view is that the R-curve will have a very low slope when the specimen is thick enough for plane-strain conditions, and that this implies negligible amounts of stable crack growth, the observations made at ARL support the view that the R-curve can still be rising enough to permit substantial amounts of stable tearing (effectively a pop-in phenomenon), often repeated many times, even when constraint conditions

approach plane strain. The implications of the presence of stable tearing need to be understood to ensure that we obtain a correct view of how close the crack is to instability.

9.4.11 Effect of Intermittent Heating on Fatigue Crack Growth (Y.C. Lam - MU)

The effects on fatigue crack growth of the application of heat treatment with and without an applied load are discussed in [43]. The investigation indicates that when heat treatment is performed simultaneously with a tensile applied load, compressive residual stress will be introduced resulting in better resistance to fatigue crack initiation and growth.

Fatigue crack growth tests were performed on single-edge-cracked plates of plain carbon-manganese steel. The fatigue life and crack rate for plates with no heat treatment and plates with heat treatment under zero stress are similar. However, for plates with heat treatment carried out under a static load equal to the maximum in the fatigue cycle, the fatigue life increased by 40% and the crack growth rate immediately after heat treatment decreased by 65%.

9.4.12 Low Cycle Fatigue of Aircraft Engines (N.S. Swansson - ARL)

An experimental programme has been undertaken to evaluate rotor component life prediction methodology by comparing the life of disc specimens under spectrum loading with life prediction estimates. An interest in application of the "Retirement for Cause" philosophy was widened to encompass the safe-life approach most commonly employed for critical rotor components. Compressor and turbine rotor disc forging blanks to an engine manufacturer's specifications were obtained as the source of disc and material test specimens. Ti-8-1-1 disc specimens with bolt-hole features were cycled to constant amplitude and military spectrum load sequences. Uniaxial bolt-hole specimens were tested to the same load spectra. Material testing of crack growth and cyclic stress-strain characteristics were performed to verify and supplement data from other sources. Moire interferometry was used to measure residual strains around bolt-hole stress concentrators and to calibrate finite element analysis [44]. Models used for fatigue life estimation include damage summation using S-N curves and local strain range, crack growth and critical crack size data.

A cyclic spin test facility for the experimental test programme has been developed. High temperature capability is incorporated and the facility has been commissioned using Incoloy 901 disc specimens. The facility can accommodate gas turbine hardware (with limitations on rotor size and test rate) and operate under service load spectra and elevated (constant) temperatures.

9.4.13 Collaborative Bonded Bush Investigation (J.M. Finney - ARL)

A collaborative test program between ARL and RAE, Farnborough is investigating the use of adhesively-bonded bushes in cracked holes to extend subsequent fatigue life. 7050-T7451 aluminium alloy specimens containing a single hole are being tested under constant-amplitude loading to grow cracks to a common length from several shapes of crack-starter notches before bonding of the stainless steel bushes. Subsequent testing will be under Falstaff loading at two stress scale levels.

9.4.14 NDE Research (D.R. Arnott - ARL)

Non-destructive evaluation research at ARL focuses primarily on ultrasonics, eddy-current and acoustic emission techniques as well as a range of optical and electrical techniques specifically directed to quality

control of adhesive bonding. One highlight from each of the two major research programs is discussed here.

(1) Ultrasonics for composite repair/reinforcement problems

Unidirectional boron/epoxy composites are used for crack repair and for reinforcement of highly stressed regions in aircraft, but there are currently no non-destructive evaluation techniques for quality control, monitoring changes in adhesive bond strength and durability, or measuring the depth of a crack under a repair patch. Theoretical and experimental research to investigate the possible use of leaky interface waves to address these problems began in 1990. The first phase of this work addresses the problem of the measurement of the elastic constants for the highly anisotropic composite. Theoretical schemes for measuring the elastic constants were developed within the restrictions of one-sided access and single orientation for the specimen. [45] and include many novel features to minimise practical measurement problems and error accumulations. Experimental studies on uni-directional boron/epoxy plates and overlays are in progress, using laser-generated ultrasonic techniques.

(2) Theoretical work to guide eddy-current inspections of advanced composite materials

Renewed interest in the problem of electromagnetic induction in anisotropic materials has followed the need to characterise the properties of carbon-fibre and metal matrix composite materials and to detect defects, such as delaminations of cracks. There is comparatively little theoretical work to guide the development of eddy-current inspection of advanced composites.

A theoretical model to describe time-harmonic electromagnetic induction in a uniaxially anisotropic plate has been derived [46]. The model assumes that the plate is non-magnetic and that the axis of anisotopy is in a plane parallel to the surface of the plate. Closed form expressions for the fields and currents induced by a general source were obtained. The general theory was illustrated by considering both pancake coil and tangent coil geometries and the change in coil impedance was calculated for an Al-B plate. The calculations were in excellent agreement with experiment.

9.4.15 NDT During F/A-18 Bulkhead Fatigue Test (G. Clark and D.R. Arnott - ARL)

Research was conducted to evaluate possible non-destructive evaluation techniques suitable for the F/A-18 FS488 bulkhead. Ultrasonic, dye penetrant, eddy-current and acoustic emission techniques were each used. Under the conditions of the test, none of the techniques reliably detected cracks prior to the failure of the bulkhead. The acoustic emission system comprised a limited six-sensor array, configured to examine highly-stressed regions below the area where failure occurred. Extensive post-processing of the recorded data indicated that the acoustic emission techniques has the potential to detect and locate cracks in the bulkhead, but only within the last 2% to 5% of the life of the bulkhead, when cracks are growing rapidly toward failure. Examination of fracture surface and crack growth rates for the failed FS488 bulkhead indicate that the safe-life of the aircraft is very sensitive to the size of small critical defects which could lead to premature crack initiation. The possibility of rogue flaws and service damage therefore causes concern for safe-life operation. Further, since unstable crack growth commences at crack depths of the order of 5 millimetres, a safety-by-inspection approach requires reliable detection of cracks at a much earlier stage of development, i.e. sub-millimetre cracks. None of the available non-destructive techniques, including acoustic emission, has yet demonstrated this capability. However, acoustic emission has demonstrated the potential to give warning of an imminent failure in the bulkhead. Research to improve confidence in the acoustic emission technique for the bulkhead application is in progress.

9.4.16 Graphite/Epoxy Environmental Exposure Programme (R. J. Chester - ARL)

Graphite/Epoxy (gr/ep) composite materials are widely employed in the structure of the F/A-18 aircraft; indeed, 34% of the external surface area and 9% of the dry structural weight is gr/ep. These materials are currently replacing conventional aluminium alloys for both major and minor components, because they offer weight reductions due to their high specific strength and stiffness and because they are generally resistant to degradation by fatigue and corrosion.

In order to support the RAAF in maintenance of the gr/ep components on the F/A-18, ARL has initiated a major programme in association with the Tropical Exposure Site at MRL-Queensland. The experiments are designed to evaluate the influence of cyclic loading and tropical exposure on a range of gr/ep specimens which represent various structural details of the aircraft; honeycomb sandwich specimens, monolithic moisture absorption coupons, adhesively bonded joints and mechanically fastened joints will all form part of the trial. The sandwich beam specimens consist of two ten-ply skins bonded to aluminium honeycomb core and as such are typical of various components on the F/A-18. These beam specimens contain either representative forms of damage (such as impact damage or teflon inclusions) or representative repairs. Constant amplitude cyclic loading at a strain level of 2500 microstrain is applied to all specimens in blocks of 100,000 cycles at a frequency of 45 cph. Between these loading blocks the beams are removed from the loading rigs and subjected to thermal cycling (-50°C to $+105^{\circ}$C and 40 overload cycles at 3500 microstrain. These conditions are designed to simulate as closely as possible the conditions of actual aircraft usage.

Damage assessment is carried out by ultrasonic C-scanning, compliance measurements and residual strength measurements at the end of the trial. The beams have experienced close to 750,000 cycles of fatigue loading in the tropical environment at the 2400 microstrain level. Compliance results and C-scans do not indicate that any significant damage growth has occurred at this stage. The strain level of these tests will soon be increased to 2750 microstrain and the tests continued in order to determine the threshold strain required for damage propagation.

In addition to these tests, exposure trials are being conducted to determine the moisture absorption characteristics of the graphite/epoxy under field conditions. An interesting result is that under full exposure to solar radiation the specimens actually lose weight despite the high ambient humidity. This is due to degradation of the surface layers of resin and to specimen heating effects which modify the local humidity environment around the specimen. Specimens have been produced with embedded thermocouples in order to obtain temperature data which will allow modelling of absorption behaviour.

9.4.17 Fatigue of Fibre Composite Materials and Components (D.S. Saunders and G. Clark - ARL)

(1) Impact damage growth in thick laminates
The research effort in the fatigue testing of thick laminates has continued at a low level over the past two years. However, existing fatigue testing specimens have been examined in an attempt to understand the mechanism(s) of damage growth in thick laminates. This work has been conducted by scanning electron microscopy of delaminated regions and optical microscopy of sections through the delamination stack. By following the progress of delaminations in the fatigue laminate it appears that, to some extent, the configuration on the initial impact damage influences the configuration of delaminations grown under fatigue loading. The present work has elucidated details of the geometrical interactions between ply cracks and delaminations in carbon fibre composite laminates after impact damage and fatigue loading. Examination of the delamination surfaces indicates that the detailed fracture mode is influenced by the presence of neighbouring delamination boundaries. Delaminations are also believed to nucleate under

fatigue loading at the cross-over points of intraply cracks: however, most of the delamination growth is associated with the delaminations formed on initial impact [47,48].

(2) Impact-damage growth monitoring

This is a continuing programme which is used to study the internal behaviour of composite laminates under fatigue loading. Any NDI techniques which provide information are considered; Moire, compliance measurement and SPATE have been used in past work.

The use of ultrasonic time-of-flight C-scanning as a method for monitoring damage growth has continued, although at a lesser level of activity than in preceding years. This method of examination has been applied to the regions surrounding fastener holes in thick laminates. It is considered that this method still provides the best information on internal damage in composite laminates.

Alternative methods for monitoring damage development in composite laminates are presently being investigated, in particular piezo-electric films acting as strain gauges over large areas of laminates and fibre-optic sensors over localised areas of laminates.

(3) Fatigue behaviour of mechanical joints in composite laminates.

The work on mechanical joints in thick composite laminates has continued and is now being extended to study the behaviour of thin laminates. The development of damage at the fastener hole in thick laminates has been studied in some detail [49]. This work has reported the investigation of wear at the fastener hole, development cracking (ply damage) around the fastener hole and delamination growth. The effects of incipient delaminations on the fatigue performance of mechanical joints in composite laminates is presently under way. Preliminary results have shown a small degradation of bearing failure loads as a result of the presence of delaminations. Fatigue testing is continuing and will include testing under hot/wet environmental conditions.

The stress-strain behaviour of mechanical joints in composite laminates has been studied for room temperature/dry and specimens conditioned under hot/wet environments and tested at room temperature. The results of this programme showed the plasticizing effect of the hot/wet environmental conditioning, but ultimate failure loads were similar.

The studies of fatigue behaviour of mechanical joints in thin laminates has only recently commenced. The effects of joint geometry on fatigue life will be studied. The testing procedures for this work have been established and for this work back-to-back testing of small coupons has been adopted.

(4) Moisture absorption by carbon fibre composite laminates

The modelling of moisture absorption by carbon fibre composite laminates is completed and no additional work is planned. Experimental studies of moisture absorption in composite laminates are continuing (Section 9.4.16).

9.4.18 Bonded Composite Repair Technology for Metallic Aircraft Components
(A.A. Baker and R. Jones - ARL)

As described in several earlier ICAF Reviews, high modulus, high strength fibre composites can be employed to provide highly efficient and cost effective reinforcements or repairs for metallic aircraft components. The composite is adhesively bonded to the component selectively in regions of high stress. This technology, pioneered at ARL since the mid-seventies, based mainly on the use of boron/epoxy composites and structural film adhesives, has saved the Royal Australian Air Force many millions of

dollars and greatly increased aircraft availability. The Defence Science and Technology Organisation (DSTO) recently signed an agreement with an Australian company, Helitech Pty Ltd, to exploit the technology on a world wide basis with the active participation with Textron Specialty Materials, USA, the manufacturer of the boron/epoxy materials. Some of our most recent applications, most of which were to demonstrate use of the repair technology on civil aircraft, are discussed below.

(1) Boron/Epoxy Reinforcement of the F-111 Wing Pivot Fitting

The main aim of the doubler depicted in Fig. 19 is to reduce the strain in a critical region of the stiffener run-out during the cold proof load test (CPLT) which is performed at the USAF facility in Sacramento. A strain reduction of over 30% is required to avoid plastic yielding in this region. Considerable effort was at devoted at ARL to this very challenging task. Full scale wing tests at ARL confirmed our design predictions that the doubler could survive the high loads and low temperatures (over 7g at - 40°C) applied during the CPLT and produce the desired 30% strain reduction. However, doubler failures occurred in early CPLTs on service aircraft. Modifications were made to the doubler system and the next aircraft successfully passed the CPLT; strain gauge measurements made on that aircraft confirmed that a strain reduction of over 30% was achieved. More recently, two further aircraft with modified doubler systems also successfully passed their CPLTs.

(2) Demonstrator Reinforcement of a Fuselage Lap Joint - Boeing 727

A boron/epoxy reinforcement approach is being developed in an attempt to overcome the problem of multi-site damage in fuselage lap joints. To validate this approach a demonstrator programme is being undertaken in conjunction with Australian Airlines, the Australian Civil Aviation Authority and the US Department of Transportation. The demonstrator programme involved the application of a reinforcement to the region of a lap-seam joint of an Australian Boeing 727 aircraft, Fig. 20. The reinforcement will be monitored during service of the aircraft to asse : bond durability; to date the reinforcement has been in service for 5000 hours with no indication of any problem.

Fatigue tests at ARL on simulated lap joints have demonstrated that substantial increases in life are obtained by the reinforcement procedure - even when the repair is exposed to severe environmental conditions.

(3) Demonstrator Reinforcement of Keel Beam - Boeing 767

Severe corrosion pitting was detected in the aluminium keel beam of an Ansett Boeing 767 aircraft. Removal of the damage resulted in reduction of thickness of the flange of the keel beam from 6.5 to 2.5 mm in several regions. In consultation with Ansett Airlines of Australia, a demonstrator boron/epoxy reinforcement was designed and developed to restore the effective stiffness of the beam. Figure 21 is a schematic diagram of the region, indicating the corrosion damage and the approximate size and location of the reinforcement. The repair region was surface treated and the reinforcement bonded with a structurel film adhesive in less than one day.

(4) Miscellaneous Demonstrator Applications - Boeing 747

In association with Qantas Airways Ltd and the Boeing Commercial Aircraft Company, simulated repairs were recently applied to a Qantas B747-300 aircraft. The purpose is to demonstrate the ease and time of application of the repairs and, subsequently, their durability. It was clearly demonstrated during the programme that the bonded repairs were much faster to apply than an equivalent metal repair. Boron/epoxy reinforcements were applied to several regions subjected to harsh environmental conditions, including: i) lower-fuselage skin, ii) trailing-edge flap, iii) engine-pylon fairing, iv) thrust-reversor cowling and v) leading-edge-nose skin.

(5) Demonstrator Repairs to Helicopter Blades

Boron/epoxy repairs were applied recently to the underside of two aluminium Bell 206L Long Ranger helicopter blades (about 1 m inboard from the blade tip) to demonstrate the types of repair that could be applied to typical damage such as a surface scratch, corrosion pit, or fatigue crack. The main aim of the programme is to test the repairs under severe erosion conditions that will arise when the blade is operated in rain. It is also important to demonstrate that the repair will be durable under high stress and severe environmental conditions. One of the helicopters is operated in the highlands of Papua New Guinea in a region of hot/wet conditions and very high rainfall. This helicopter has experienced 600 hours flying time with no evidence to date of erosion damage to the patch or of any disbond problems. The other helicopter is operating in a marine environment in Victoria, Australia. To date, no problems have been experienced, but this trial is at an even earlier stage.

9.5 REVIEW OF STRUCTURAL FATIGUE RESEARCH AND DEVELOPMENT IN NEW ZEALAND

9.5.1 Andover C. Mk. 1 (A.D. James and W.L. Price - DSE)

Cracks have been found in the wing lower rear spar booms of a number of RNZAF Andover medium transport aircraft. The defects were located at an attachment point for an element of the main landing gear retraction system. Analysis of the defects confirmed that they were fatigue cracks and that the crack growth rates were relatively low. The primary loading action responsible for crack growth was associated with undercarriage extension and retraction.

Work has also started on the initial phases of a Loads/Environment Spectrum Survey (L/ESS) on an RNZAF Andover. The project, which employs microprocessor strain recorder equipment, is aimed at comparing RNZAF use of the aircraft with the design fatigue spectrum. The Andover airframe is similar to that of the HS 748 aircraft and the L/ESS data will also be compared to the HS 748 full scale fatigue test spectrum.

9.5.2 P-3K Orion (P.C. Conor - DSE)

The New Zealand version of the P-3B aircraft has been in service for approximately 25 years. A recent assessment of counting accelerometer data has indicated that the recent use of the aircraft has been comparatively severe. The airframes, which contain a significant number of corrosion damage repairs, are consequently considered likely to have accumulated a relatively high amount of fatigue damage. The discovery in late 1990 of a fatigue crack at a wing spar repair in one aircraft has confirmed that fatigue damage rates in service have been high.

As a result, an L/ESS programme has been initiated to assess the RNZAF operational usage spectrum in detail and to provide data which can be compared with information provided by other P-3 operators.

9.5.3 UH-1H Iroquois Helicopter (A.D. James - DSE)

In early 1990, an RNZAF Iroquois helicopter was found to have come close to losing its tailfin and tail rotor after a large fatigue crack developed in the fin spar. Inspection of other helicopters in the fleet showed that a high proportion of these were cracked in the same location. The defects had initiated at a fastener hole at the intersection between the fin. and tailboom. Fracture analysis indicated that although the average crack growth rates were likely to have been low, bursts of accelerated fatigue crack propagation had occurred from time to time.

An evaluation of local strain levels in flight showed that strain maxima occurred during high speed flying. The installation of a stainless steel doubler repair is likely to solve the immediate problem but the area will continue to be inspected frequently because an analysis has shown that strain levels in the fin spar may still be comparatively high.

9.5.4   Aermacchi MB 339 C (P.C. Conor - DSE)

The introduction of the MB 339 C jet trainer into RNZAF service has presented an opportunity to monitor fatigue damage accumulated by individual structural components from zero flying hours. The RNZAF has purchased the Aermacchi Airborne Strain Counter (ASC) system with the new aircraft and the equipment is to be used to gather information to meet individual aircraft tracking objectives. An operational strain survey is also expected to be carried out to supplement the rainflow count data supplied by the ASC equipment.

9.5.5   Fatigue Analysis (P.C. Conor - DSE)

A high degree of empiricism is involved in the calculation of crack growth behaviour in metal components. Even recently-developed crack propagation prediction models still require extensive testing and adjustment. Not only is this process extremely costly for the operators of small fleets of aircraft, but significant uncertainties are introduced if the aircraft load spectrum deviates from that assumed when the crack growth calculations were originally performed.

The development of more refined models is likely to depend on the achievement of an improved understanding of the interactions between fatigue cracks and the materials they are growing through. Research work is therefore being aimed at the evaluation of the small-scale crack acceleration and retardation effects which occur under spectrum loading. A high resolution DC potential drop crack growth measurement system has been developed for the task and this system, together with computer-based data acquisition and control equipment is being employed to assess incremental crack growth rates in response to variable amplitude loads. In parallel with the experimental work, a Dugdale strip-yield crack closure model is being used to provide analytical predictions which can be compared against the experimental data.

9.6   BIBLIOGRAPHY ON THE FATIGUE OF MATERIALS, COMPONENTS AND STRUCTURES

The fourth Volume of this work by J.Y. Mann covering the period 1966 to 1969 was published by Pergamon Press, England early in 1990. It contains 5,709 references on the subject and, together with the first three Volumes of the Bibliography covering the periods 1838-1950, 1951-1960 and 1961-1965, provides a total of 21,075 references.

Volume 4 was prepared directly from references stored on a personal computer. References for subsequent years are being progressively entered into this data base. During the last two years the emphasis has been on the decade 1970-1979 and, in particular, the years 1970-1973 which will be the period covered by Volume 5.

There are now over 23,000 entries on disc in the period 1970-1986 inclusive, of which over 6,000 are in the four-year period 1970-1973. It is estimated that the decade 1970-1979 will ultimately include over 20,000 citations, and that by the year 1974 a figure of over 2,000 references per annum will have been achieved.

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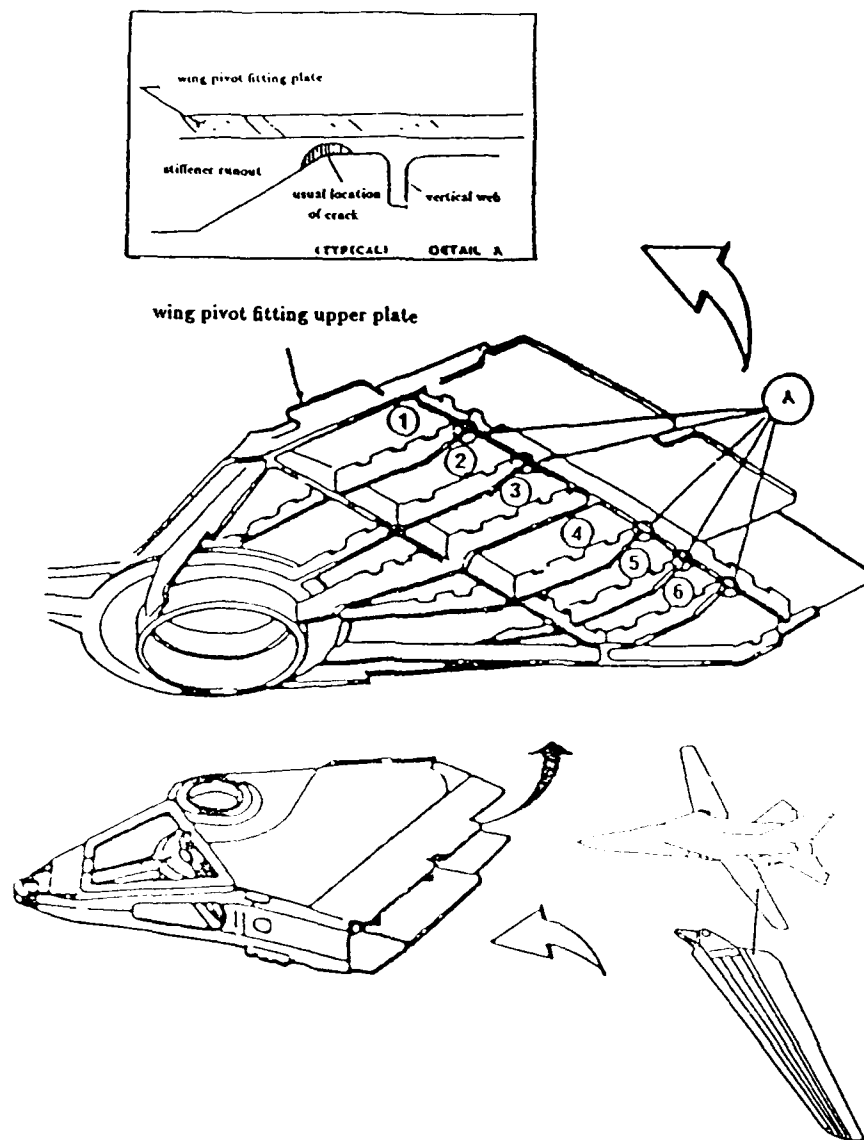
FIG. 1   F-111 WING PIVOT FITTING - LOCATION OF CRITICAL REGION ON STIFFENER RUNOUT

FIG. 2   NOMAD FULL SCALE FATIGUE TEST - SCHEMATIC OF STUBWING BY-PASS STRUCTURE
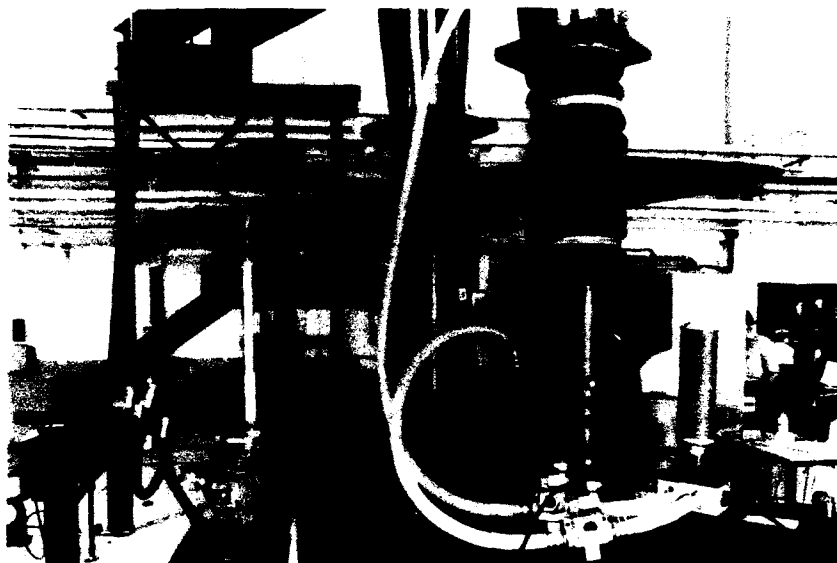
FIG. 3   DEMONSTRATOR F/A-18 AIRBAG MANOEUVRE LOADING SYSTEM
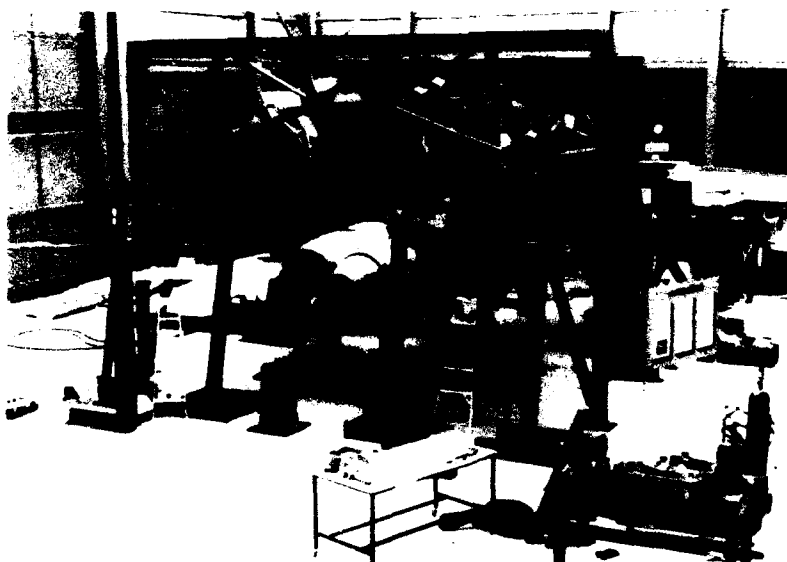
FIG. 4   F/A-18 TEST ARTICLE AND LOADING FRAMEWORK

FS 488
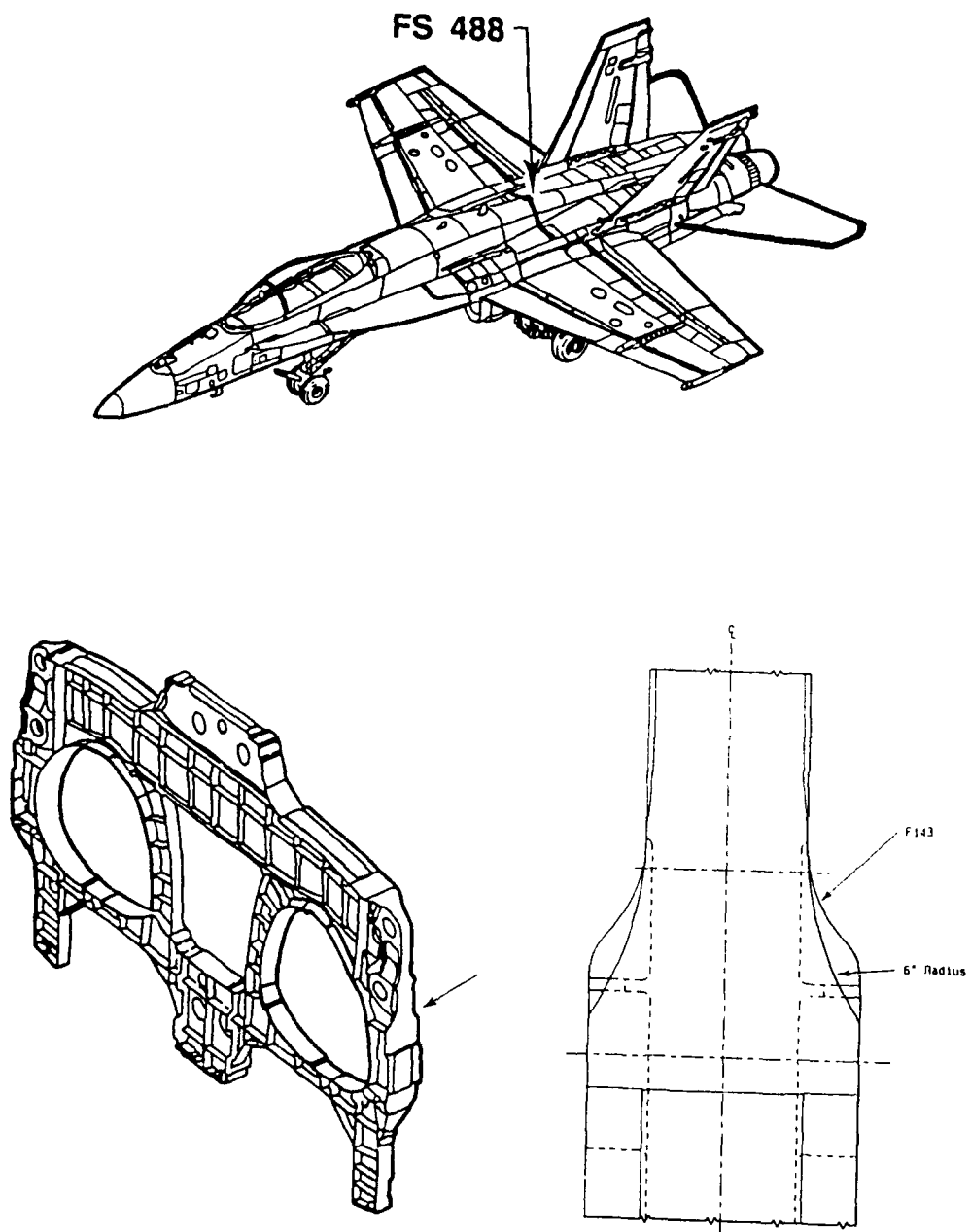
F143

6" Radius

FIG. 5   FUSELAGE STATION 488 BULKHEAD SHOWING EARLIER (F143) AND

LATER (6" RADIUS) FLANGE CONFIGURATIONS
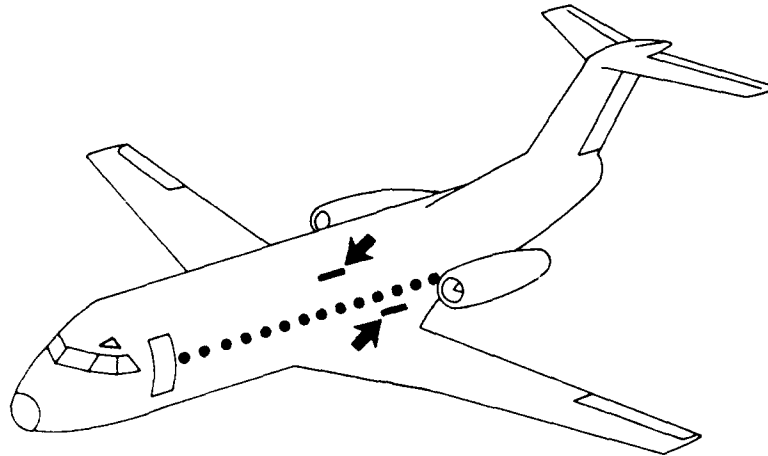
FIG. 6   FOKKER F28 SHOWING LOCATION OF FUSELAGE LAP JOINT CRACKING
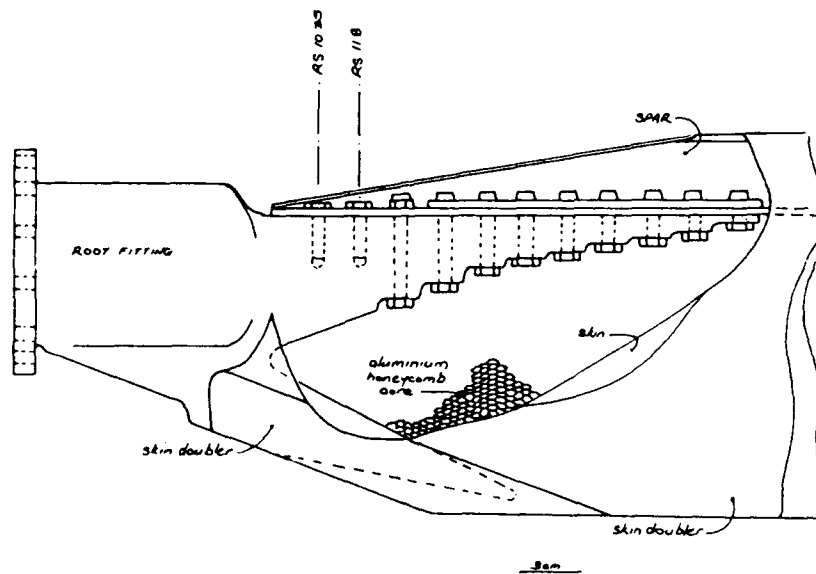
FIG. 7   ROBINSON R-22 MAIN ROTOR BLADE CONSTRUCTION DETAIL
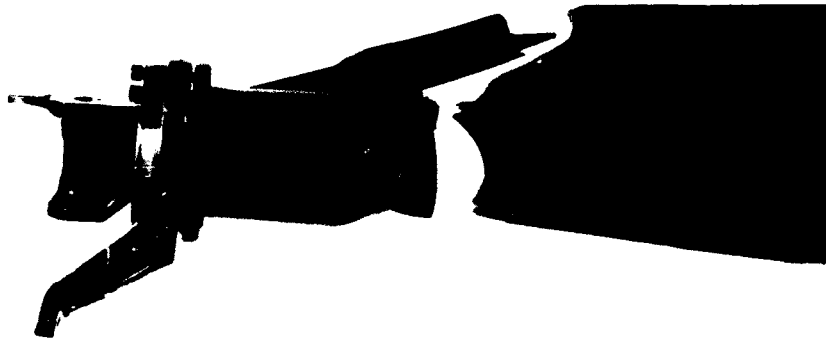
FIG. 8   ROBINSON R-22 MAIN ROTOR BLADE FAILURE
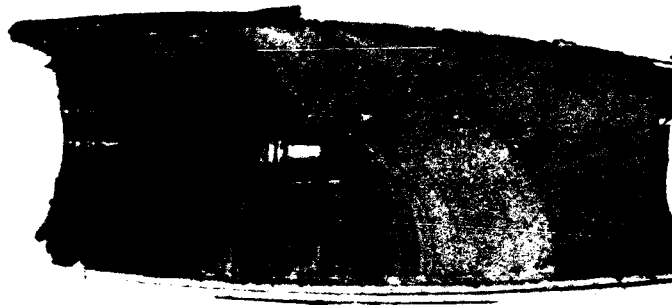
FIG. 9   ROBINSON R-22 MAIN ROTOR BLADE FAILURE FRACTURE SURFACE
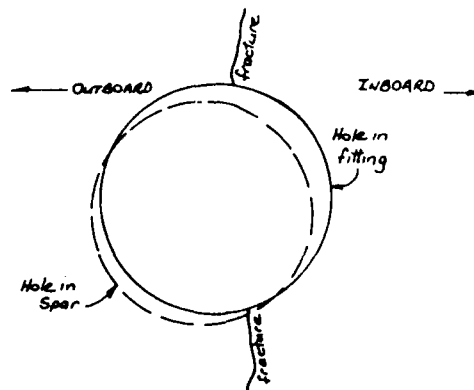
FIG. 10   ROBINSON R-22 MAIN ROTOR BLADE FAILURE

HOLE MISMATCH AT FAILURE LOCATION
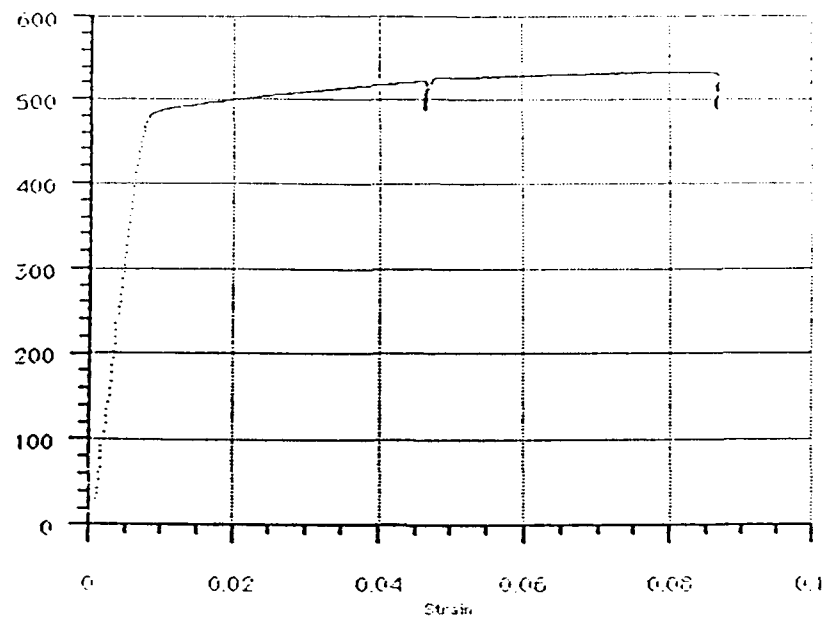
FIG. 11   EXPERIMENTAL STRESS-STRAIN CURVE WITH SHORT (10 MIN) STRAIN
HOLDS AT STRAINS OF 0.045 AND 0.085
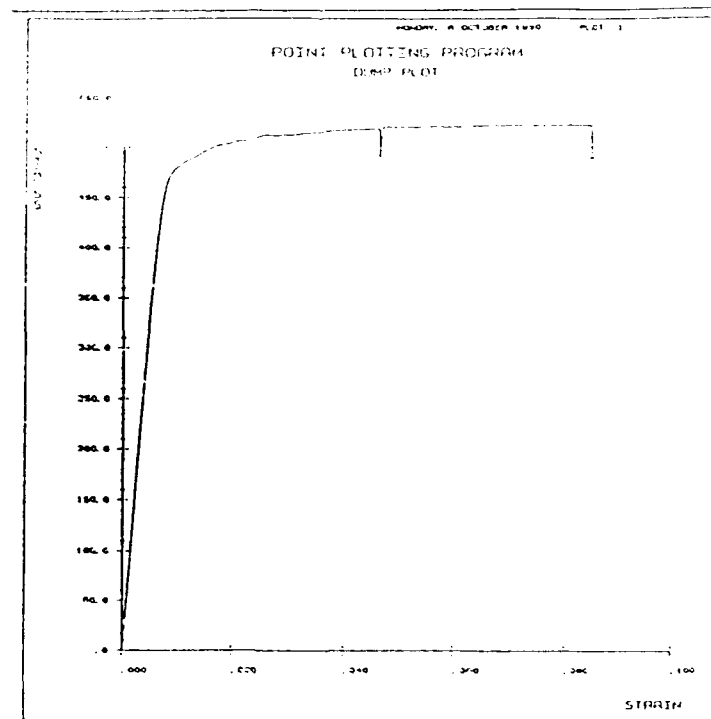
FIG. 12   MODEL RESULTS FOR STRESS-STRAIN CURVE WITH SHORT (10 MIN)
STRAIN HOLDS AT STRAINS OF 0.045 AND 0.085
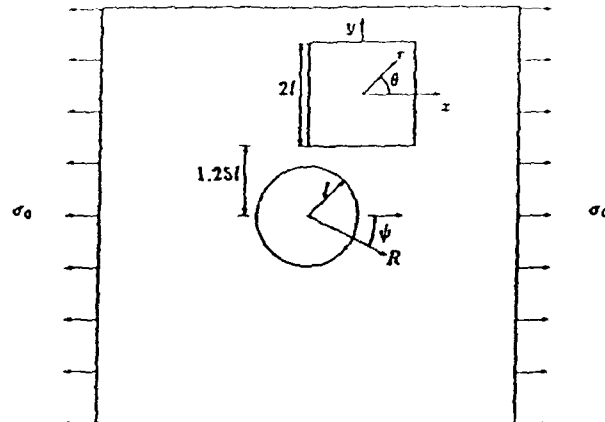
FIG. 13   GEOMETRY OF CIRCULAR HOLE IN UNIAXIALLY LOADED SQUARE
PLATE SHOWING SQUARE SCAN REGION $\sigma_0 = 50$
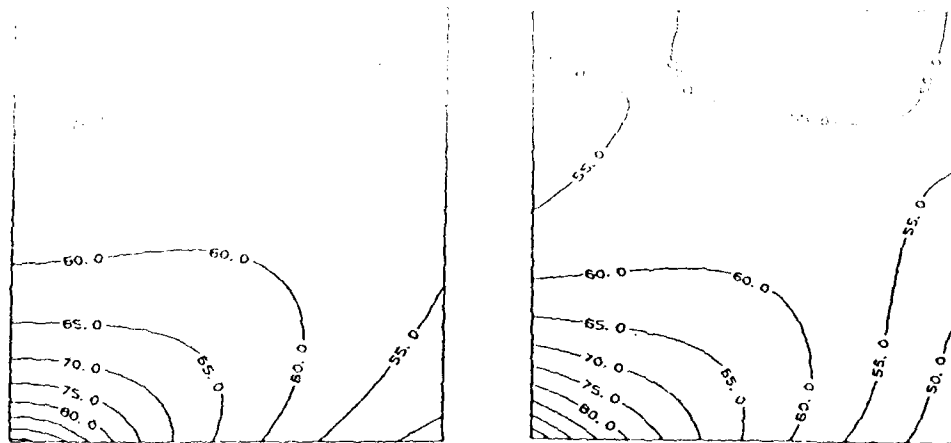
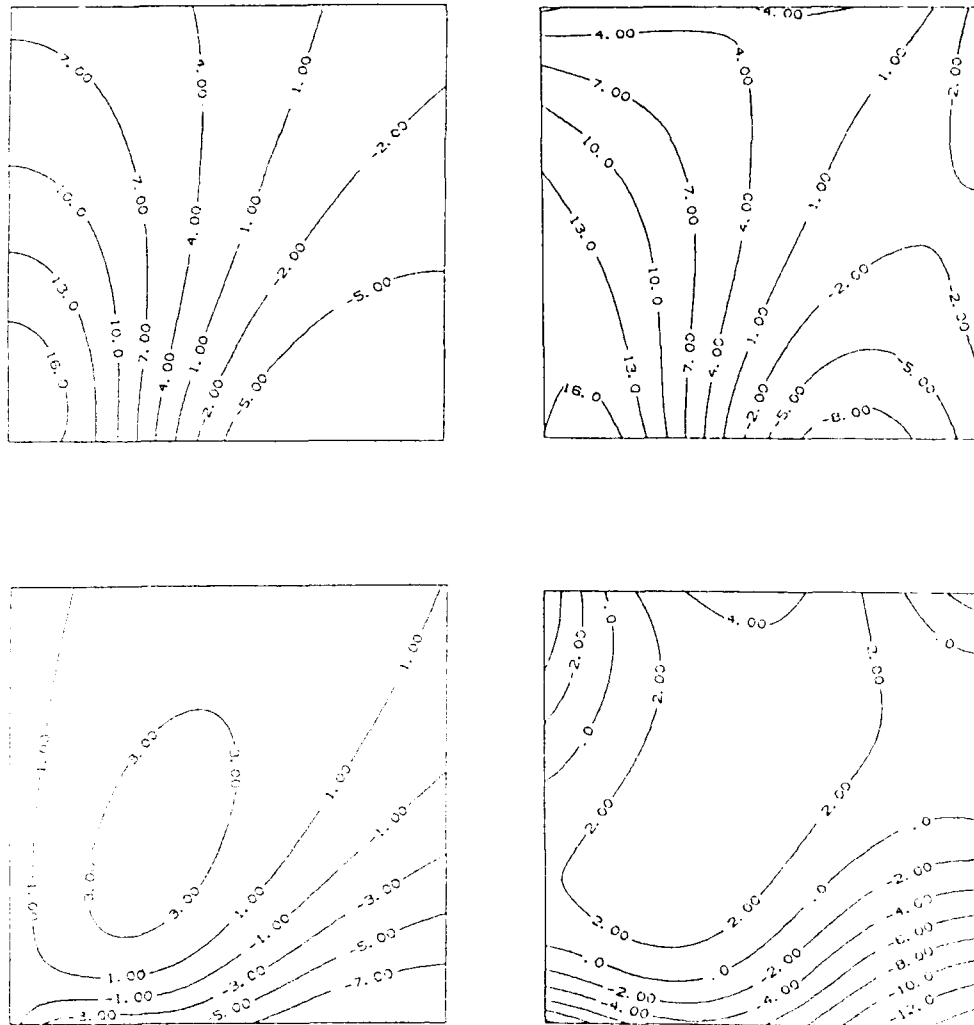FIG. 14   COMPARISON OF DIRECT (LEFT) AND INVERSE (RIGHT) σ_{xx} STRESSES
FOR GEOMETRY OF FIG. 13

FIG. 15   COMPARISON OF DIRECT (LEFT) AND INVERSE (RIGHT) STRESSES FOR
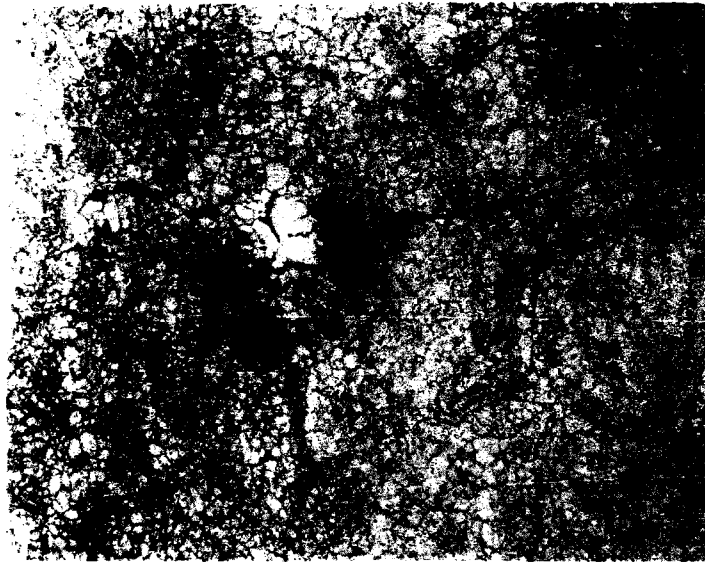GEOMETRY OF FIG. 13. UPPER: σ_{yy}, LOWER: τ_{xy}

FIG. 16   TYPICAL BULKHEAD DEFECT WITH SEVERAL FATIGUE CRACKS
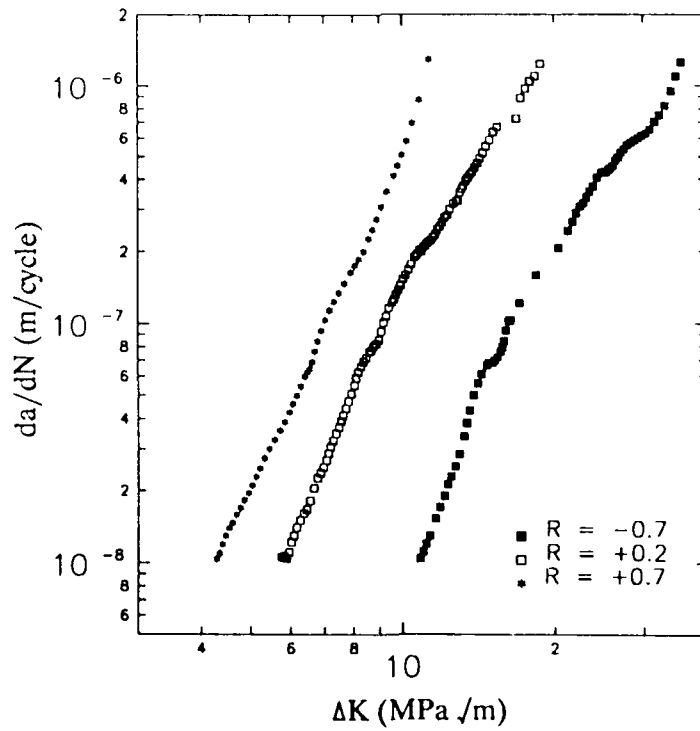GROWING FROM INTERGRANULAR PENETRATIONS

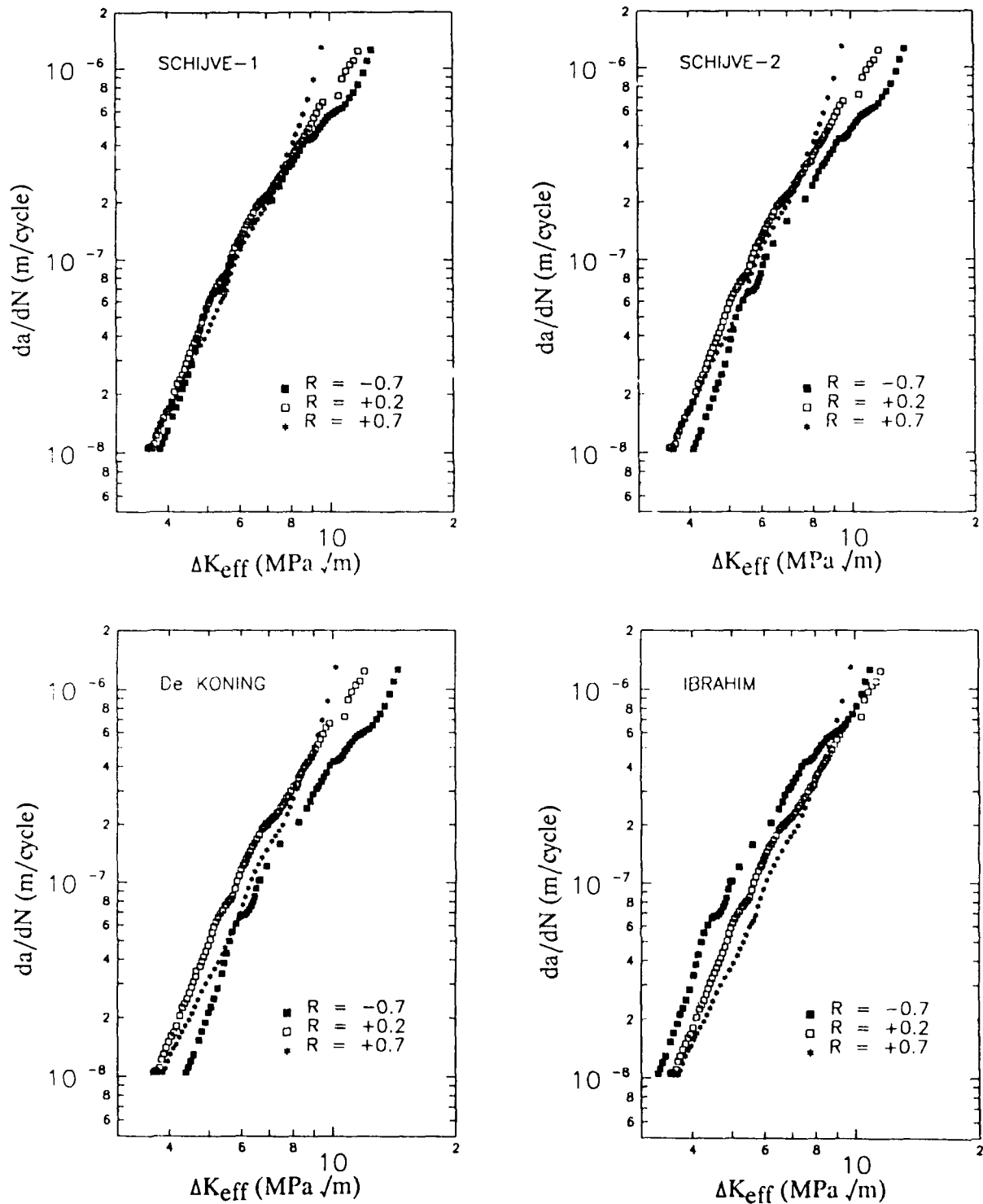FIG. 17   CRACK GROWTH RATE DATA FOR A7-U4SG-T651 (2214-T651) ALUMINIUM ALLOY

FIG. 18   CRACK GROWTH RATE DATA OF FIG. 17 PLOTTED AS ΔK$_{eff}$ USING FOUR
DIFFERENT EQUATIONS
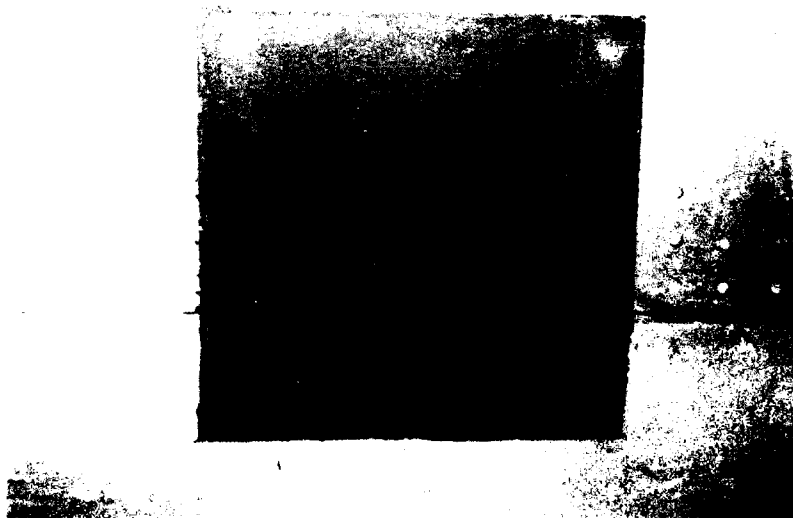
FIG. 19   F-111 WING PIVOT FITTING DOUBLERS

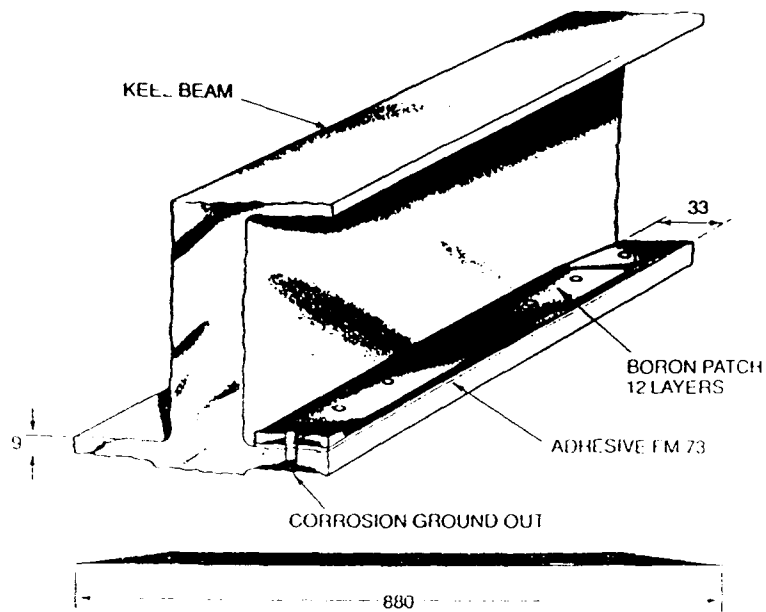FIG. 20   727 FUSELAGE LAP JOINT DOUBLER

KEEL BEAM

33

BORON PATCH
12 LAYERS

ADHESIVE FM 73

9

CORROSION GROUND OUT

880

FIG. 21   767 KEEL BEAM DOUBLER

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16. ABSTRACT

This document was prepared for presentation to the 22nd Conference of the International Committee on Aeronautical Fatigue scheduled to be held in Tokyo, Japan on May 20 and 21, 1991.

A review is given of the aircraft fatigue research and associated activities which form part of the programmes of the Aeronautical Research Laboratory, Universities, the Civil Aviation Authority, the Australian aircraft industry and the Defence Scientific Establishment, New Zealand. The major topics discussed include fatigue of both civil and military aircraft structures, fatigue damage detection, analysis and repair and fatigue life monitoring and assessment.

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